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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

REPORT 922

CHARACTERISTICS OF LOW-ASPECT-RATIO WINGS AT SUPERCRITICAL MACH NUMBERS

By JOHN STACK and W. F. LINDSEY

1949

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AERONAUTIC SYMBOLS

1. FUNDAMENTAL AND DERIVED UNITS

	Symbol	Metric		English	
		Unit	Abbreviation	Unit	Abbreviation
Length	l	meter	m	foot (or mile)	ft (or mi)
Time	t	second	s	second (or hour)	sec (or hr)
Force	F	weight of 1 kilogram	kg	weight of 1 pound	lb
Power	P	horsepower (metric)	-------	horsepower	hp
Speed	V	kilometers per hour	kph	miles per hour	mph
		meters per second	mps	feet per second	fps

2. GENERAL SYMBOLS

W Weight $=mg$

g Standard acceleration of gravity $=9.80665$ m/s² or 32.1740 ft/sec²

m Mass $=\frac{W}{g}$

I Moment of inertia $=mk^2$. (Indicate axis of radius of gyration k by proper subscript.)

μ Coefficient of viscosity

ν Kinematic viscosity

ρ Density (mass per unit volume)

Standard density of dry air, 0.12497 kg-m^{-4}-s^2 at 15° C and 760 mm; or 0.002378 lb-ft^{-4} sec^2

Specific weight of "standard" air, 1.2255 kg/m³ or 0.07651 lb/cu ft

3. AERODYNAMIC SYMBOLS

S Area

S_w Area of wing

G Gap

b Span

c Chord

A Aspect ratio, $\frac{b^2}{S}$

V True air speed

q Dynamic pressure, $\frac{1}{2}\rho V^2$

L Lift, absolute coefficient $C_L=\frac{L}{qS}$

D Drag, absolute coefficient $C_D=\frac{D}{qS}$

D_0 Profile drag, absolute coefficient $C_{D_0}=\frac{D_0}{qS}$

D_i Induced drag, absolute coefficient $C_{D_i}=\frac{D_i}{qS}$

D_p Parasite drag, absolute coefficient $C_{D_p}=\frac{D_p}{qS}$

C Cross-wind force, absolute coefficient $C_C=\frac{C}{qS}$

i_w Angle of setting of wings (relative to thrust line)

i_t Angle of stabilizer setting (relative to thrust line)

Q Resultant moment

Ω Resultant angular velocity

R Reynolds number, $\rho\frac{Vl}{\mu}$ where l is a linear dimension (e.g., for an airfoil of 1.0 ft chord, 100 mph, standard pressure at 15° C, the corresponding Reynolds number is 935,400; or for an airfoil of 1.0 m chord, 100 mps, the corresponding Reynolds number is 6,865,000)

α Angle of attack

ϵ Angle of downwash

α_0 Angle of attack, infinite aspect ratio

α_i Angle of attack, induced

α_a Angle of attack, absolute (measured from zero-lift position)

γ Flight-path angle

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By JOHN STACK and W. F. LINDSEY

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Langley Air Force Base, Va.

National Advisory Committee for Aeronautics

Headquarters, 1724 F Street NW., Washington 25, D. C.

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CHARACTERISTICS OF LOW-ASPECT-RATIO WINGS AT SUPERCRITICAL MACH NUMBERS

By John Stack and W. F. Lindsey

SUMMARY

The separation of the flow over wings precipitated by the compression shock that forms as speeds are increased into the supercritical Mach number range has imposed serious difficulties in the improvement of aircraft performance. These difficulties arise principally as a consequence of the rapid drag rise and the loss of lift that causes serious stability changes when the wing shock-stalls. Favorable relieving effects due to the three-dimensional flow around the tips were obtained and these effects were of such magnitude that it is indicated that low-aspect-ratio wings offer a possible solution of the problems encountered.

(NACA abstract) (S[illegible] 1866)

INTRODUCTION

Flight at supercritical Mach numbers has appeared extremely difficult because of rapid drag increase and marked stability and control changes. The change in stability has been found in many instances to be so great as to cause loss of normal control of the aircraft. Serious buffeting of the tail usually accompanies these adverse stability changes.

The adverse effects are shown in reference 1 and elsewhere to be directly connected with a change of flow over the wing. This change of flow is precipitated by the formation of an essentially normal shock, which produces separation of the flow over the wing. The separated flow, as now seems clear, was indicated in reference 2 to be an outstanding contributing cause of the drag rise. The stability change encountered with airplanes is largely a consequence of either or both the change in angle of zero lift of the wing or the change in lift-curve slope of the wing when the separated flow occurs. Elimination of the separated flow could be expected to alleviate to a large extent the difficulties encountered.

Elimination of the separated flow could possibly be accomplished by boundary-layer control, though experiments made thus far indicate alleviation, but not elimination, of the separated condition. Other difficulties both aerodynamic and structural are, however, encountered. Because it appears clear that the normal-shock phenomenon produces the separation of the flow, some modification to reduce the shock losses could be expected to contribute markedly toward solution of the present difficulties.

Unpublished results of experimental investigations of the flow around simulated propeller tips in the Langley 11-inch and 24-inch high-speed tunnels showed marked delay and alleviation in the adverse effects at supercritical Mach numbers as compared with results obtained in two-dimensional flows. It was likewise shown that the shocks formed at and near the tips were not normal to the stream. These results are substantiated by the work of reference 3 performed on actual rotating propellers. These results suggest the existence of marked three-dimensional relieving effects at tip sections of wings or propellers. Wings of low aspect ratio could therefore be expected to undergo much less adverse effects at supercritical Mach numbers than wings of present conventional aspect ratios.

Considerations of the effects of aspect-ratio reduction indicate that other effects may be expected. Thus, the slope of the lift curve is determined by the infinite aspect ratio or section characteristics plus the induced effects. If the induced angle, as with a low-aspect-ratio wing, is large, a given change in section characteristics should produce smaller relative change in lift-curve slope than would occur with a wing of moderate or high aspect ratio for which the induced angle is small. Further, reduction of wing aspect ratio through increasing the downwash angle at the tail reduces the stabilizing effect of the tail until finally a value of aspect ratio is reached for which the geometrical and the downwash angles are approximately equal. When this condition is reached, changes in the flow over the tail as a result of changes in wing characteristics may not produce large changes in stability. The stability will then depend primarily on the wing characteristics.

As a consequence of the foregoing considerations, experiments were conducted in 1944 in the Langley 24-inch high-speed tunnel to study the characteristicss of low-aspect-ratio wings at supercritical Mach numbers. The experiments reported herein consisted of tests of wings of aspect ratios A ranging from ∞ to 2. All the wings were of NACA 0012 section. The tips were cut square, each aspect ratio being obtained by progressively cutting the tips off the original wing. The speed range extended to Mach numbers exceeding 0.9.

APPARATUS AND TESTS

The investigation was conducted in the Langley 24-inch high-speed tunnel, which is a nonreturn induction-type tunnel (reference 2). The induction nozzle, located downstream from the test section, induces the air to flow from the atmosphere through the tunnel. The length of air passage from the region of low-velocity air at the entrance section to the test section is small, approximately 4 feet (fig. 1). The absence of a return passage, the short entrance length, and the strong favorable pressure gradient along most of the entrance length provide a very thin boundary layer along the walls of the test section.

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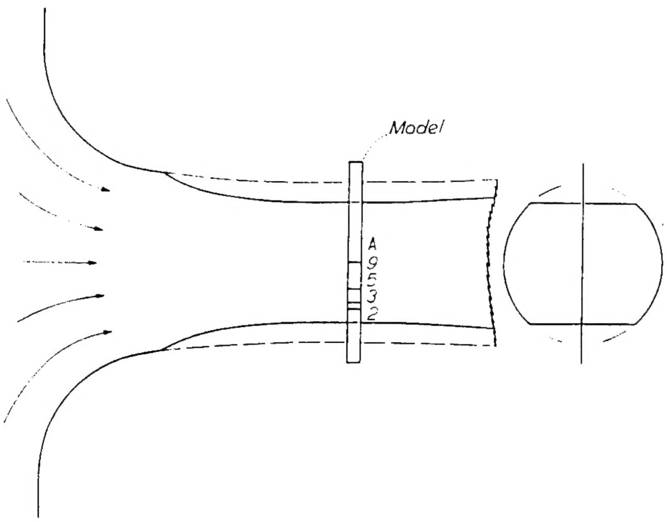


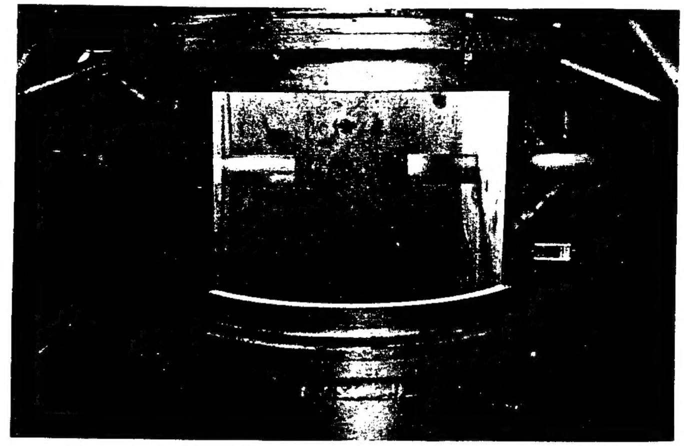
FIGURE 1.—Langley 24-inch high-speed tunnel. Schematic diagram illustrating tunnel shape and models.

The test section, originally circular (24-in. diameter), had been modified prior to the present investigation by the installation of flats on the tunnel walls. These flats reduced the width of the tunnel at the test section from 24 inches to 18 inches and changed the shape of the test section from circular to one more nearly approaching rectangular. The cross section of the tunnel at the model location is shown in figure 1.

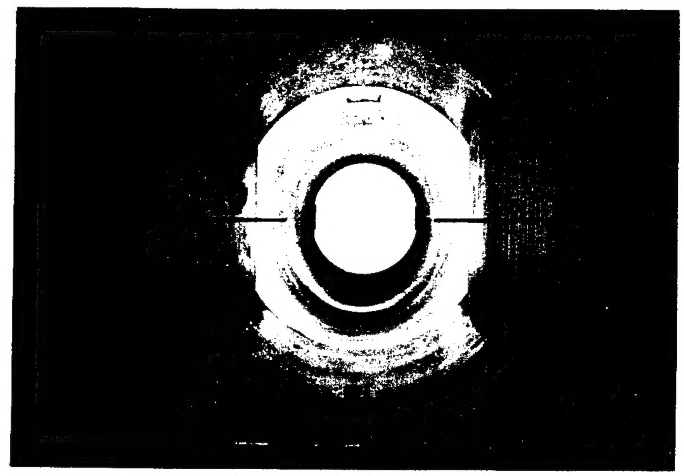
The model for the infinite-aspect-ratio tests completely spanned the test section and passed through end plates fitted into the flat walls of the tunnel. The end plates, which accurately preserved the contours of the tunnel wall at the intersection of the tunnel wall and the model, provided clearances between model and tunnel wall and thereby permitted the forces acting on the model to be transmitted to and recorded by a 3-component balance to which the ends of the model were attached (fig. 2).

The models for tests of finite aspect ratios were installed and supported in the same manner as for tests of infinite aspect ratios except that the models extended one semispan into the air stream from the tunnel wall. This type of installation is satisfactory because the boundary layer on the tunnel wall, as previously discussed, is very thin. For the tests of wings having aspect ratios of 5 or less, two semispans, one from each wall, were installed for the purpose of doubling the magnitude of the forces to be measured by the standard balance of this tunnel, which was designed for larger force ranges than those encountered in these tests. For the wings of aspect ratios 5 and 7, tests were made with one and two semispan models mounted in the tunnel. The results of tests made with both one and two semispans mounted in the tunnel were in close agreement even without the tunnel-wall corrections.

Lift, drag, and pitching moment were measured on wings having rectangular plan forms and zero twist. The aspect



(a) Over-all view with access door removed, showing model installation.



(b) Downstream view with model in place.

FIGURE 2.—Model mounting in test section of Langley 24-inch high-speed tunnel. Aspect ratio, 5; two-semispan installation.

ratios of the wings tested were ∞, 9, 7, 5, 3, and 2. All the wings were of the NACA 0012 profile and had chords of 2 inches. Tests were made at angles of attack from 0° to 6° and at Mach numbers between 0.5 and the tunnel choked condition, that is, the maximum Mach number obtainable for a given model-tunnel combination. The Reynolds number range corresponding to this Mach number range is from 5.3×10^5 to 7.6×10^5.

The various factors affecting the accuracy of these data may, in general, be divided into two classes: accidental errors and systematic errors.

The accidental errors arose from inaccuracies in the calibrations of the balance and the static-pressure orifices and from design limitations on the maximum sensitivity of the balance. The maximum sensitivity appears to be the primary source of accidental errors and is a maximum for the small-area wing of aspect ratio 2 at low Mach numbers. At a Mach number of 0.50 for the wing of aspect ratio 2, the accidental errors in coefficients appear to be of the following order:

Wing lift coefficient, C_L	± 0.008
Wing drag coefficient, C_D	± 0.0010
Wing pitching-moment coefficient, $C_{m_{c/4}}$	± 0.010

The systematic errors arose from end interference and tunnel-wall interference. The data have been corrected for end interference resulting from the small leakage through the clearance gap at the juncture of the model and the tunnel wall by corrections determined experimentally (reference 4). With the wings of low aspect ratio, a large part of the whole span is affected by the gap leakage, and the effects of the leakage are therefore expected to be relatively greater. It is believed that the drag coefficients for the low-aspect-ratio wings are higher and the lift-curve slopes are lower than would be found in the absence of leakage. These effects are being studied experimentally in the Langley 24-inch high-speed tunnel.

Tunnel-wall interference has been investigated theoretically (reference 5) and the existence of constriction effects in high-speed tunnels has been shown experimentally (references 6 and 7). The theoretically derived corrections, however, have not been experimentally verified at supercritical Mach numbers. The errors indicated theoretically by the method of reference 5 increase as model size, Mach number, drag coefficient, and lift coefficient increase. The theoretically indicated errors for the infinite-aspect-ratio wing at an angle of attack of 6° and a Mach number M of 0.84 are:

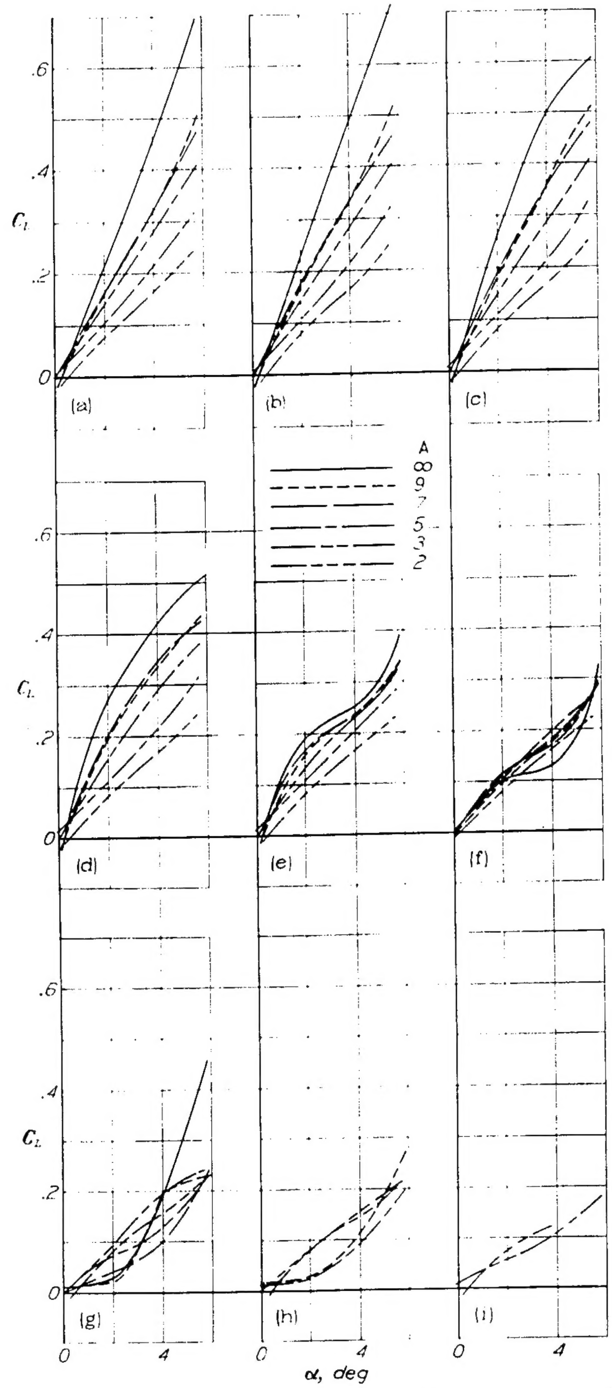
$$\text{Corrected } M = M \times 1.014$$
$$\text{Corrected } C_L = C_L \times 0.982$$
$$\text{Corrected } C_D = C_D \times 0.980$$

These values indicate that the effect of tunnel-wall interference on these data is small, 2 percent or less, and therefore no correction has been applied.

The choking phenomenon is an additional factor that enters into the problem of testing at high Mach numbers. At the choked Mach number sonic velocities extend from model to tunnel wall and the static pressure is lower behind the model than it is ahead; thus large gradients in the pressure are produced (reference 7). The resulting flow past the model is unlike any free-air condition. Data obtained at the choked Mach number are therefore of questionable value and are not presented herein.

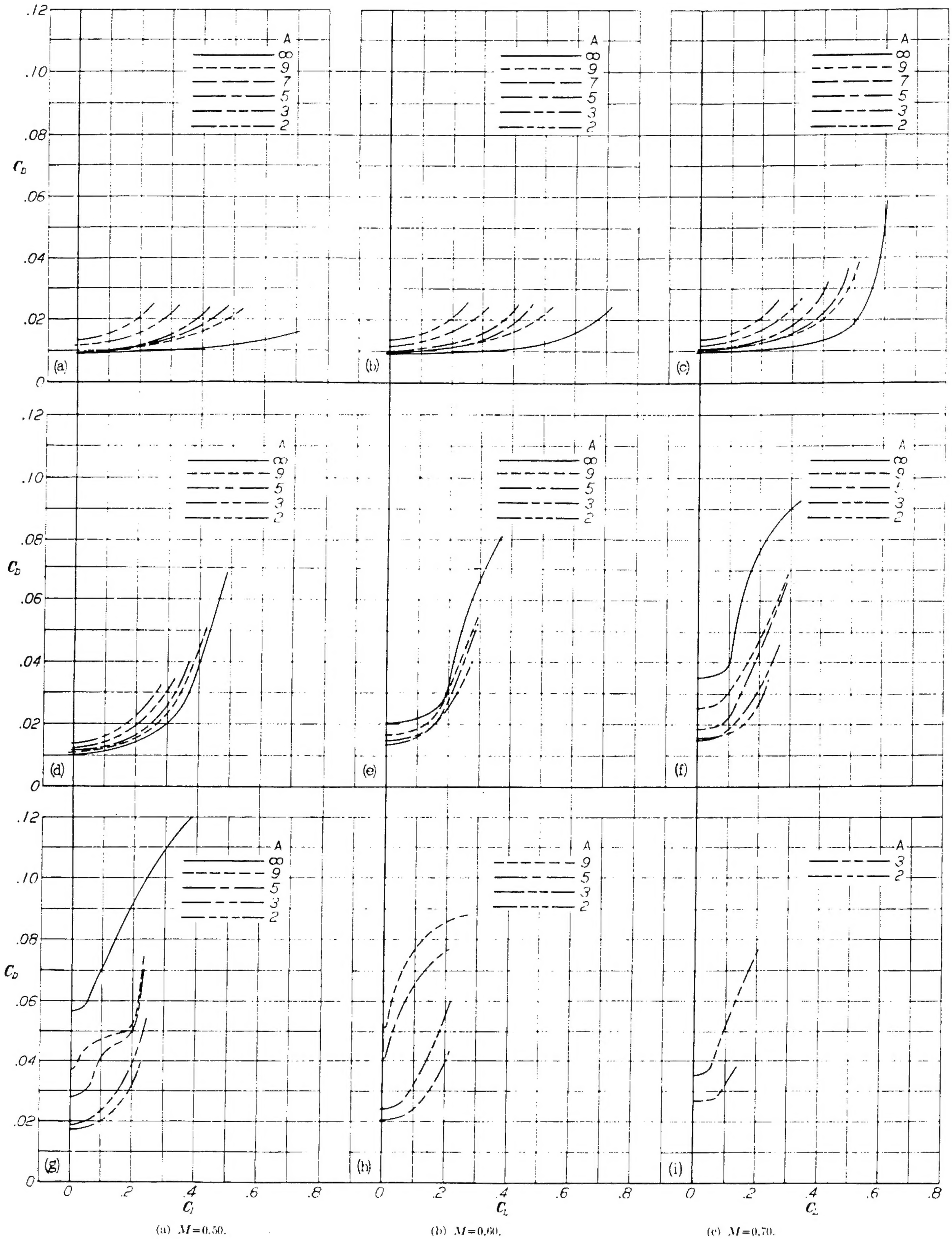
RESULTS

The basic results are presented in figures 3 to 5. Figure 3 shows the lift coefficient C_L plotted against the angle of attack α for each of nine values of the Mach number from 0.5 to 0.9. Similarly, the drag results are shown by polar diagrams in figure 4. The moment coefficients are given in figure 5 plotted against lift coefficient for six values of the Mach number in the range from 0.5 to 0.9. The minimum drag coefficients for the various aspect ratios are shown in figure 6. All the results are for the actual aspect ratios tested and are not corrected to infinite aspect ratio. Thus, the induced effects are included. Since the wings of higher aspect ratio have the lower choking Mach numbers, data for these wings are presented for somewhat lower Mach numbers than the data for the lower aspect ratios. As noted previously, no data at the choked condition are presented. The highest Mach number for which data are presented for each wing is approximately 0.025 less than the corresponding choking Mach number.



(a) $M=0.50$. (b) $M=0.60$. (c) $M=0.70$.
(d) $M=0.75$. (e) $M=0.80$. (f) $M=0.825$.
(g) $M=0.85$. (h) $M=0.875$. (i) $M=0.90$.

FIGURE 3.—Lift curves for various aspect ratios and Mach numbers. NACA 0012 section; rectangular plan form and tip.



(a) $M=0.50$.
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(h) $M=0.875$.

(c) $M=0.70$.
(f) $M=0.825$.
(i) $M=0.90$.

FIGURE 4.—Polars for various aspect ratios and Mach numbers. NACA 0012 section; rectangular plan form and tip.

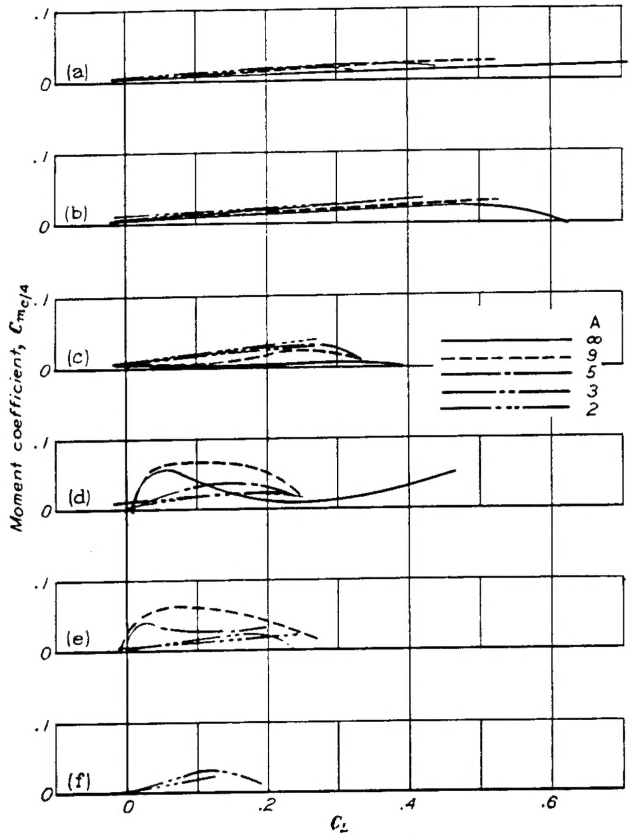

FIGURE 5.—Moment-lift curves for various aspect ratios and Mach numbers. NACA 0012 section; rectangular plan form and tip.

DISCUSSION

Figures 3 and 4 show a most pronounced change in finite wing characteristics as the critical Mach number of the sections, approximately 0.72 for the NACA 0012 sections used herein, is exceeded. For conventional and higher aspect ratios, the lift curves (fig. 3) show irregularities in slope and effectively discontinuous slope changes at supercritical speeds. These irregularities, which are the principal cause of the stability difficulties that have been encountered at supercritical speed, appear first at the higher lift coefficients encountered in the pull-out condition, but as the speed is increased they occur at progressively lower lift coefficients until finally irregularities occur in the low-lift region around zero lift coefficient. For the higher-aspect-ratio wings tested in the present investigation, the lift-curve slope decreases almost to zero in the low angle-of-attack range at Mach numbers between 0.85 and 0.875.

The low-aspect-ratio wings (aspect ratios 2 and 3), however, show none of the characteristic lift-curve irregularities at the high Mach numbers. The lift-curve slopes for the low-aspect-ratio wings also show relatively little change with Mach number. The usual rise of lift-curve slope with Mach number through the subcritical speed range is absent as is the abrupt fall in slope at supercritical speeds. A partial explanation for the absence of the increase of lift-curve slope with Mach number in the subcritical range is given by considering the finite-wing characteristics to be composed of the infinite-wing or section characteristics and the induced characteristics. The induced characteristics are determined principally by the lift coefficient and are, in first-order approximation, independent of the Mach number. Hence, when the induced characteristics are large, as for the low-aspect-ratio wings, a given change in section characteristics produces less relative change of lift-curve slope than is usually expected or obtained for wings of high or conventional aspect ratios for which the induced characteristics are relatively small.

The effects of aspect ratio on the drag characteristics as shown by the polar diagrams (fig. 4) indicate very marked departure from the usual low-speed characteristics when the speeds are increased to supercritical values. The results presented in figure 4, as previously noted, include the induced drag. At the lower speeds the low-aspect-ratio wings have the highest drag, as could be determined by theory. As the critical speed of the basic section of the wings is exceeded, however, the differences in drag diminish and the polar curves approach coincidence (fig. 4 (d)). With still further increase of speed, the order of the variation of drag with aspect ratio reverses; the low-aspect-ratio wings, even including the induced drag as in figure 4, have markedly reduced drag as compared with the high-aspect-ratio wings. This change in characteristics is associated with delayed and less rapid rise of drag as the aspect ratio is decreased. Both the delayed drag rise and the slower rate of drag rise are illustrated for the minimum drag attitude (0° angle of attack and zero lift for the symmetrical section) in figure 6. The section critical Mach number is given in the figure for comparison. For the wings of aspect ratios 2 and 3, the Mach number for significant drag rise is approximately 0.1 higher than for the infinite- or high-aspect-ratio wings and the initial rate of drag rise is much less.

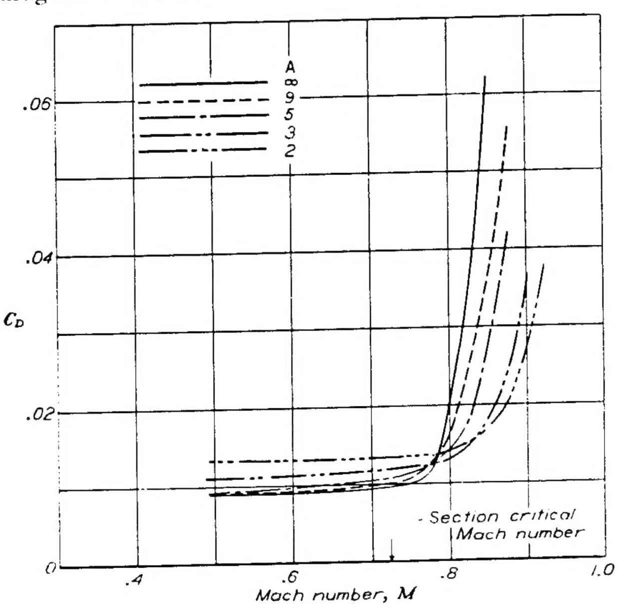

FIGURE 6.—Variation with Mach number of the minimum drag coefficient for wings of various aspect ratios.

The pitching-moment-coefficient results, though not so complete as the lift-coefficient and drag-coefficient data, show changes as the speed is increased from subcritical to supercritical values that are in general of the same character as the lift-coefficient and drag-coefficient changes. At subcritical Mach numbers the aspect ratio has but little influence on the values of the moment about the wing quarter-chord axis. The slope of the curve of moment against lift is slightly positive, indicating that for the section used, the aerodynamic center is slightly ahead of the wing quarter-chord axis. For all Mach numbers up to 0.7, as indicated by figures 5 (a) and 5 (b), the aspect ratio has but little influence on the moment coefficient for a given lift coefficient. Increase of Mach number up to the critical value causes, in accordance with the known theories, a small increase in the moment coefficient. When large supercritical values of the Mach number are reached, however, drastic changes in the wing quarter-chord moment coefficient are found for the high-aspect-ratio wings. Abrupt changes in the variation of the moment coefficient with the lift coefficient occur at very low lift coefficients and at higher lift coefficients these curves tend to give a stable slope as distinguished from the usual unstable slope characteristics of low-speed data. This change in moment characteristics has been shown previously and is due to the movement of the shock on the wing.

Reduction of aspect ratio reduces the changes in moment coefficient, as is shown in figure 5 by the data for wings of aspect ratios 2 and 3. The slope remains positive for all Mach numbers up to 0.9, and the change in moment coefficient from the values obtained at or near the wing-section critical Mach number (0.72) as compared with the changes for the wings of conventional aspect ratios is relatively small up to the highest speeds investigated.

The over-all effects of reducing the aspect ratio on improving the undesirable wing characteristics are very great. The absence of irregularities in the lift curve, the indicated freedom of the lift curve from drastic slope changes, and the similar effects for the wing moment curve indicate that the serious stability changes which have occurred with conventional aircraft when flown in the supercritical region may be alleviated in large degree. Likewise the delayed drag rise and the less rapid rate of drag increase at the high supercritical Mach numbers permit increased speed.

The improved supercritical-speed characteristics found for the low-aspect-ratio wings are a consequence of the three-dimensional type of flow at the tip. Because the effects of the flow at the tip are quite large, the tip shape is likely to be of great importance. In the present experiments the tip shape was made square principally as a matter of convenience in using an existing model to investigate the over-all effect. The square tip leads to large local velocities and at low speeds is known to produce undesirable disturbances. It is likely, therefore, that an appreciable improvement in the low-aspect-ratio-wing characteristics may be obtained by suitably shaping the tip. Large local velocities that would occur over the forward and middle parts of the tip can lead to large disturbances, probably involving shock, which might produce at least partly separated flows over the rear part of the wing. General considerations of the flow about the tip indicate that a change of plan form from the square type used in these illustrative experiments to a tapered plan form giving reduced chord at the tip and a rounded or elliptical tip shape may produce further favorable effects. Likewise, a thinner section of late critical Mach number type can be expected to delay the onset of the drag rise until much higher speeds. These experiments indicate that the serious adverse compressibility phenomena, particularly as regards drag and lift, are delayed to Mach numbers exceeding 0.9 by a low-aspect-ratio wing of rectangular plan form with a conventional 12-percent-thick section. Use of a 10-percent-thick wing of late-critical-speed type will, on the basis of two-dimensional data for the wing sections, give a further rise of 0.08 in the critical Mach number. This change together with an improved tip shape and plan form appears to offer a new possibility of overcoming the existing problems of flight in the transonic speed range.

Though not specifically shown by the present results, two other advantages are offered by the low-aspect-ratio wing. First, thin sections giving high critical speeds may be used without the usually imposed condition of inadequate wing depth for an efficient structure, and second, the spanwise center-of-pressure shift in the supercritical region will be much reduced because of the short spanwise length.

CONCLUSIONS

The detrimental effects of compressibility in the supercritical speed range on the stability and performance of aircraft are alleviated to very great degree by the use of low-aspect-ratio lifting surfaces. Further consideration of the advantages of this type of configuration is warranted.

Langley Memorial Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Langley Field, Va., *September 6, 1945.*

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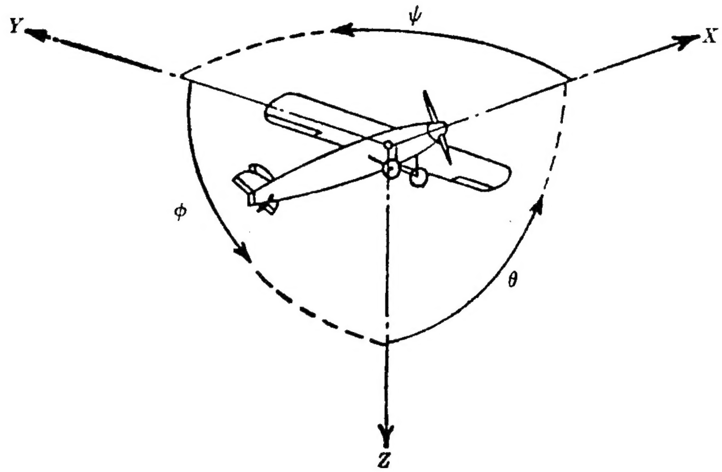

Positive directions of axes and angles (forces and moments) are shown by arrows

Axis		Force (parallel to axis) symbol	Moment about axis			Angle		Velocities	
Designation	Symbol		Designation	Symbol	Positive direction	Designation	Symbol	Linear (component along axis)	Angular
Longitudinal	X	X	Rolling	L	$Y \longrightarrow Z$	Roll	ϕ	u	p
Lateral	Y	Y	Pitching	M	$Z \longrightarrow X$	Pitch	θ	v	q
Normal	Z	Z	Yawing	N	$X \longrightarrow Y$	Yaw	ψ	w	r

Absolute coefficients of moment

$C_l=\frac{L}{qbS}$ (rolling) $C_m=\frac{M}{qcS}$ (pitching) $C_n=\frac{N}{qbS}$ (yawing)

Angle of set of control surface (relative to neutral position), δ. (Indicate surface by proper subscript.)

4. PROPELLER SYMBOLS

- D Diameter
- p Geometric pitch
- p/D Pitch ratio
- V' Inflow velocity
- V_s Slipstream velocity
- T Thrust, absolute coefficient $C_T=\frac{T}{\rho n^2 D^4}$
- Q Torque, absolute coefficient $C_Q=\frac{Q}{\rho n^2 D^5}$
- P Power, absolute coefficient $C_P=\frac{P}{\rho n^3 D^5}$
- C_s Speed-power coefficient $=\sqrt[5]{\frac{\rho V^5}{P n^2}}$
- η Efficiency
- n Revolutions per second, rps
- Φ Effective helix angle $=\tan^{-1}\left(\frac{V}{2\pi r n}\right)$

5. NUMERICAL RELATIONS

1 hp=76.04 kg-m/s=550 ft-lb/sec
1 metric horsepower=0.9863 hp
1 mph=0.4470 mps
1 mps=2.2369 mph

1 lb=0.4536 kg
1 kg=2.2046 lb
1 mi=1,609.35 m=5,280 ft
1 m=3.2808 ft